MARS RECONNAISSANCE ORBITER

PRELIMINARY ENVIRONMENTAL REQUIREMENTS AND ESTIMATES

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SECTION 1
INTRODUCTION

1.1 PURPOSE

This document defines the expected environments in which Mars Reconnaissance Orbiter flight hardware is expected to encounter, survive, and operate. These are the environments which are established to serve as a basis for deriving the final, detailed environmental design and test requirements for the Orbiter system and for each assembly, including Payloads.

1.2 SCOPE

The document includes Orbiter system and assembly environments from ground handling through Mars orbit. 'Orbiter' in this context is defined as the Orbiter bus plus the science Payloads/experiments. Where different environments apply to the Orbiter bus vs the Payloads, these will be noted.

1.3 CONTRACTOR DERIVED ENVIRONMENTS

The Orbiter contractor/partner shall prepare, with JPL consultation and approval, and document a final set of design and test environments, with appropriate margins (ref Appendix A), that are dependent on the launch vehicle selected, Orbiter configuration, and environments to be encountered through science and relay missions. Payload suppliers will be provided by the MRO Project Office a set of environmental test requirements based upon this document and Orbiter interface environments.

3.1 APPLICABILITY

These environments are applicable to both the MRO Orbiter bus and Payloads and to all flight assemblies including GFE, subcontractor, and contractor supplied.

1.5 APPLICABLE DOCUMENTS

The following documents, of the latest issue in effect, form a part of this document to the extent specified herein:

NASA-STD-7001	Payload Vibroacoustics Test Criteria
NASA-STD-7003	Pyroshock Test Criteria
JPL D-17868, Rev 1	Design, Verification/Validation and Operations Principles for Flight Systems
MIL-STD-461	Electromagnetic Emission and Susceptibility Requirements for the Control of Electromagnetic Interference
MIL-STD-462	Electromagnetic Emission and Susceptibility, Test Methods for
JPL D-20381	MRO Orbiter Requirements
JPL D-20327	MRO Mission Assurance Requirements
JPL D-560	JPL Standard for System Safety
JPL D-14040	Process and Technical Guidelines for Spacecraft Hardware Project-Specific Environmental Assurance
JPL 900-434	Standard Environmental Testing Facilities and Practices

SECTION 2
GROUND OPERATIONS AND HANDLING ENVIRONMENT

The ground operations and handling estimates encompass the environments which the Orbiter hardware may encounter during assembly level fabrication, integration, calibration, and pre-launch operations. The ground handling environments also include transportation and storage of the hardware in shipping containers.

2.1 SHIPPING AND TRANSPORTATION DYNAMICS

Shipping containers and transportation procedures for flight hardware shall be designed such that shipping and transportation dynamic environments are less severe than expected launch and flight environments.

2.2 TEMPERATURE, PRESSURE AND HUMIDITY

Temperature, pressure and humidity for ground operations such as assembly and system level testing, launch site operations, transportation or storage shall be maintained within limits that do not adversely affect flight hardware. If the environments are not controlled then environmental estimates more extreme than flight environments are required.

2.2.1 Controlled Environment

A controlled environment exists when the temperature, pressure and humidity ranges are maintained within specified limits. Table 2-1 presents the controlled thermal environment for Mars Reconnaissance Orbiter hardware. For operation of flight hardware in an environment at the upper limit of controlled temperature range, the hardware shall be monitored to assure that Protoflight test limits are not exceeded. Mars Reconnaissance Orbiter hardware shall be monitored for temperature and humidity during ground transportation.

Table 2-1. Controlled Thermal Environment for Ground Handling, Transportation and Storage

Control Parameter	Low Limit	High Limit
Temperature	$+5^{\circ}C$	$+45^{\circ}C$ (1)
Temperature Change Rate	$-5^{\circ}C/hr$	$+5^{\circ}C/hr$
Pressure	7×10^4 N/m^2 (520 torr)	1×10^5 N/m^2 (760 torr)
Relative Humidity	$\geq 30\%$ (2)	$\leq 70\%$

NOTES: (1) If flight hardware is to be operated within $10^{\circ}C$ of this limit, the hardware should be monitored to ensure that its Protoflight test level is not exceeded.

(2) Could be as low as 0% relative humidity during shipping or storage.

2.2.2 Uncontrolled Thermal Environment

Table 2-2 presents the uncontrolled thermal environmental ranges for all climates and handling altitudes. If the assemblies are designed to accommodate or is expected to be exposed to the uncontrolled environments of Table 2-2, then design verification tests at these levels are mandatory. Otherwise, protective containers with active control shall be provided.

Table 2-2 Uncontrolled Thermal Environment for Ground Handling, Transportation and Storage

Control Parameter	Low Limit	High Limit
Temperature	$-40^{O}C$	$+70^{O}C$ (1)
Temperature Change Rate	$-15^{O}C/hr$	$+15^{O}C/hr$
Pressure	1.2×10^{4} N/m^{2} (87.5 torr)	1×10^{5} N/m^{2} (760 torr)
Relative Humidity	$\geq 0\%$	$\leq 100\%$

2.3 ELECTROMAGNETIC COMPATIBILITY (EMC)

The EMC environment will be derived from unique MRO EMC environments, MIL-STD-461 and –462, the Delta III Payload Planners Guide, or other selected launch vehicle specifications, and the launch site RF environments. Testing shall include the following:

(1) power and signal line conducted susceptibility (CS);
(2) power and signal line conducted emissions (CE);
(3) radiated susceptibility (RS);
(4) radiated emissions (RE);
(5) grounding and electrical isolation (elimination of signal and power line ground loops).

Flight hardware shall be compatible with MIL-STD-461C Part 3, Class A2a, Conducted and Radiated Emissions and Susceptibility.

The EMC values of Table 2-3 are applicable design and test requirements, inclusive of margins, for flight assemblies and reflect ground, launch vehicle, and Orbiter (including payload elements) generated disturbances throughout all mission phases.

Payload-Generated EMC Fields

A radar sounder Payload, if flown on MRO, could produce electric fields on the order of 100 (TBR) Volts per meter (V/m) at frequencies 20 MHz ± 5 MHz (TBR). The 100 V/m field estimate (TBR - worst case anywhere on the surface of the Orbiter) assumes that the radar Payload uses two 3.5-meter (TBR) dipole segments as the emitter. The mechanical configuration of the Orbiter is assumed to keep the radar emitter not less than one (1) meter from other Payloads and external Orbiter mounted assemblies. As a minimum, all Payloads and Orbiter mounted assemblies shall, in a power-off state, survive exposure to the operating radar Payload without damage.

Table 2-3 EMC Requirements

Physical Characteristics	Values
Conducted Emissions Spikes, DC Power	MIL-STD-461C Part 3 CE07. Amplitude = ½ Line Voltage
Conducted Emissions Power Quality: Ripple and Noise	CE01 and CE03 shall be measured on the Proposer's equipment and shall comply with the limits in Figure 2-1
Radiated Emissions Electric Fields Narrowband	Unintentional radiated narrowband electric field levels produced shall not exceed the levels specified in Figure 2-2
Radiated Susceptibility	The Proposer's equipment shall operate without degradation of performance when subjected to the electric field strengths shown in Figure 2-3.
Conducted Susceptibility Switching Transient	MIL-STD-461C Part 3 method CS06. The switching transient shall be ± 1x line voltage, line to line or line to chassis.
Conducted Susceptibility Narrowband Ripple	MIL-STD-461C methods CS01 and CS02 as tailored in Figure 2-4.

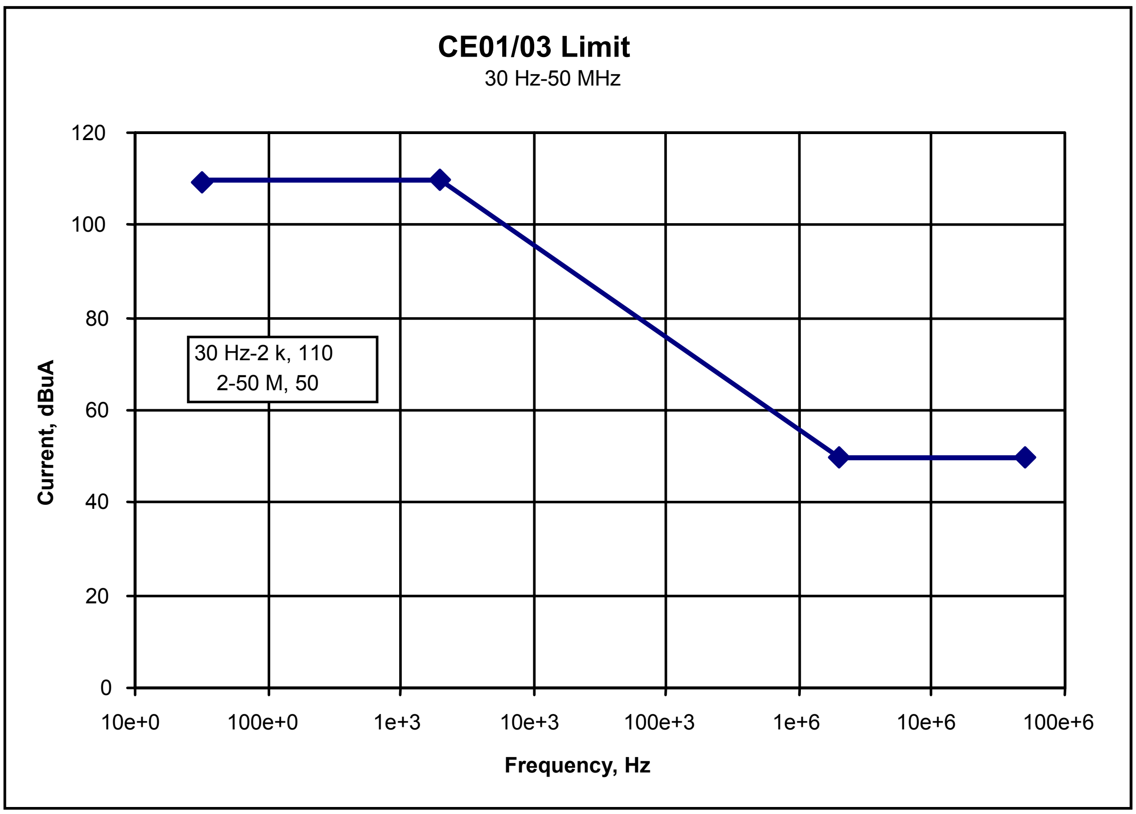


Figure 2-1 CE01/CE03 Limit – Narrowband

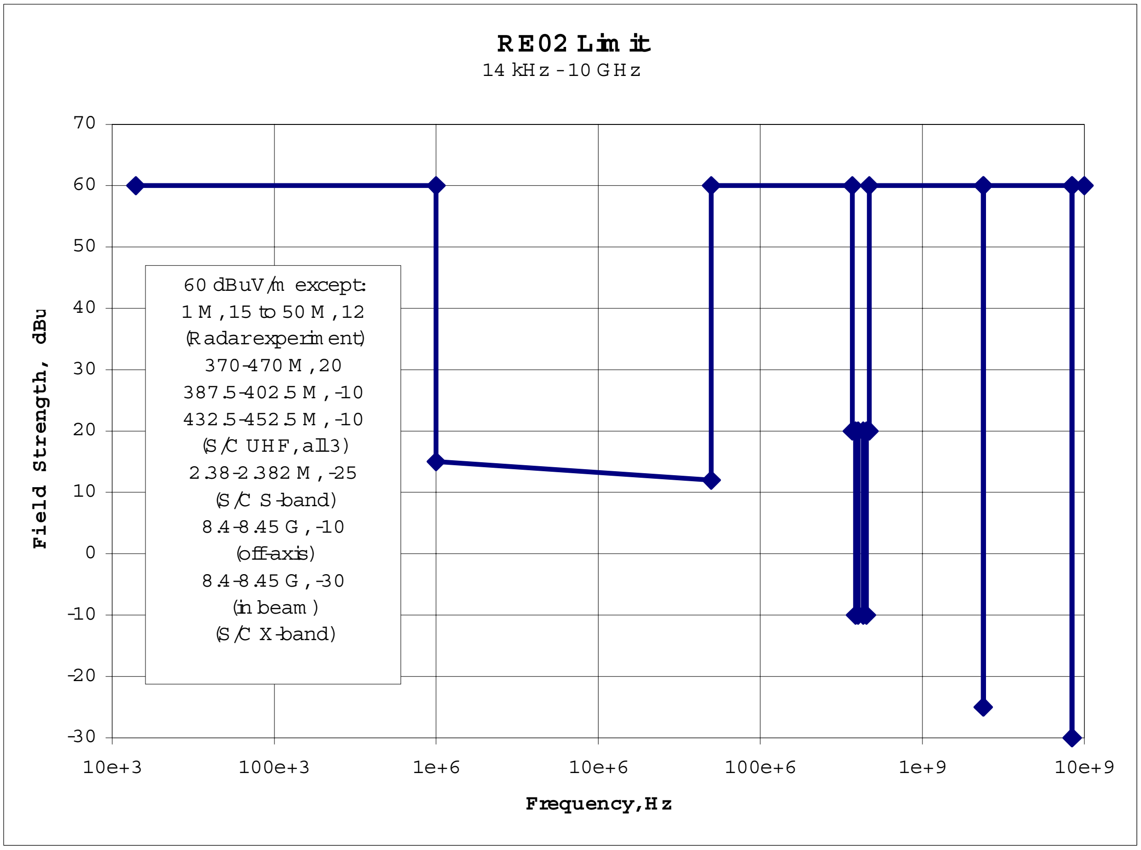


Figure 2-2 RE02 Limit – Narrowband

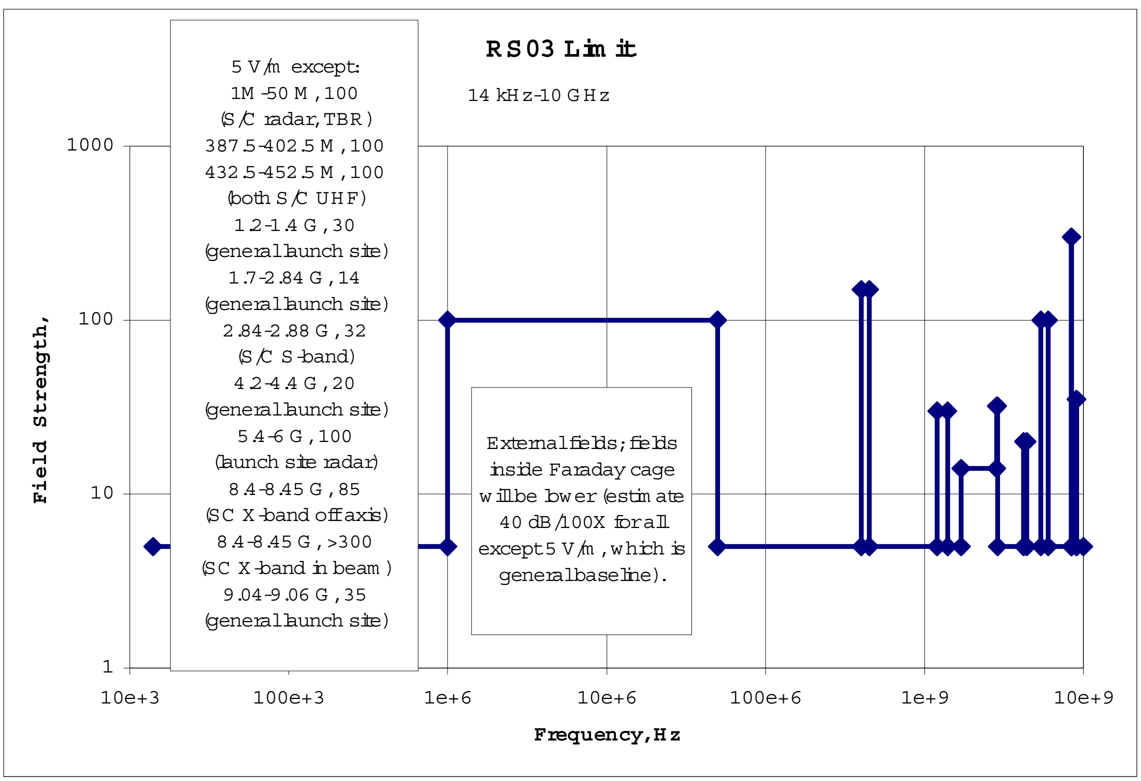


Figure 2-3 RS03 Radiated Susceptibility Limits

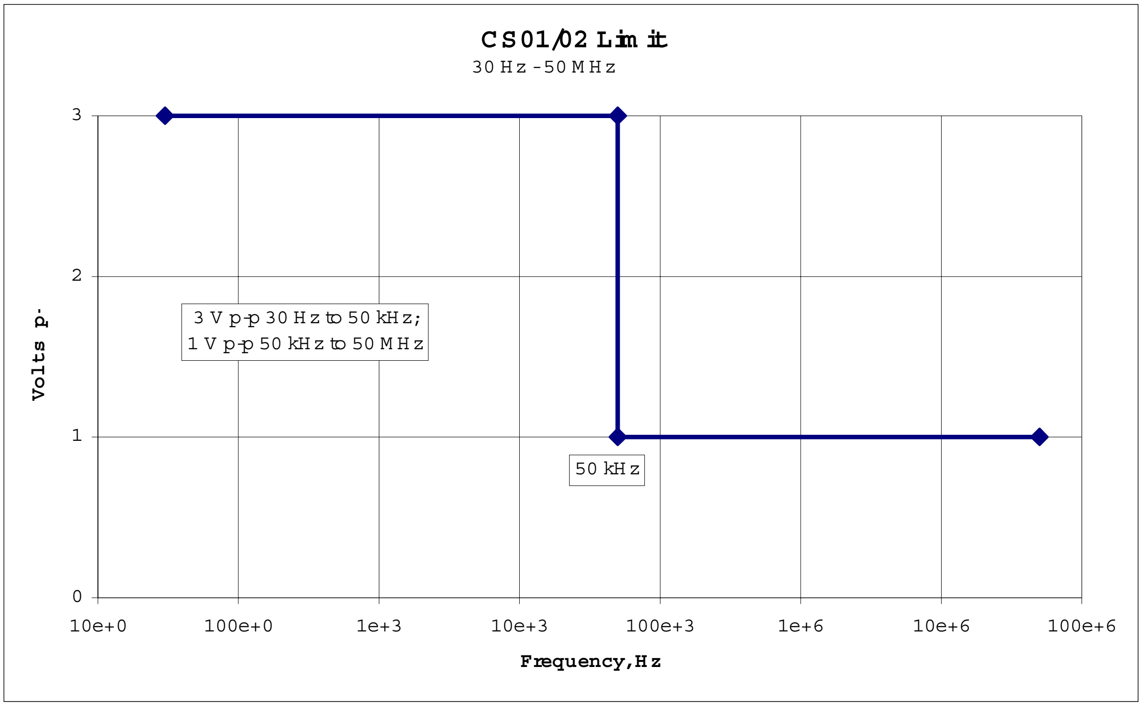


Figure 2-4 CS01 / CS02 Narrowband Injected Ripple

SECTION 3
LAUNCH ENVIRONMENTS

The launch environments are those that the Mars Reconnaissance Orbiter are expected to encounter during pre-launch operations, launch and ascent modes. The Orbiter launch environments envelope those of Delta IV, Atlas IIIB, and Atlas V, as contained in their respective payload planner's guides. Environments to be considered in those documents include but are not limited to: dynamics excitations (acoustics, transient accelerations [engine ignitions and stagings], random vibrations, and pyro shock events [fairing jettison and Orbiter/upper stage separation]), thermal environments (on-pad air conditioning, fairing temperature, free molecular heating, sun attitude); EMC/EMI; explosive atmosphere; pressure rate of change; vacuum conditions; rotational speed: etc.

3.1 DYNAMIC EXCITATIONS

3.1.1 Acoustics

The maximum acoustic environment experienced by the Orbiter occurs during liftoff and the transonic/maximum dynamic pressure flight regime. The overall sound pressure level (OASPL) envelope for the candidate launch vehicles is 141dB expected for maximum flight level and 144 dB for Protoflight test, as shown in the Figure 3-1 and Table 3-1 acoustics spectra. The protoflight acoustics test duration is 60 seconds. Acoustics testing will be performed at the Orbiter level and, in selected cases, at the Payload or assembly level.

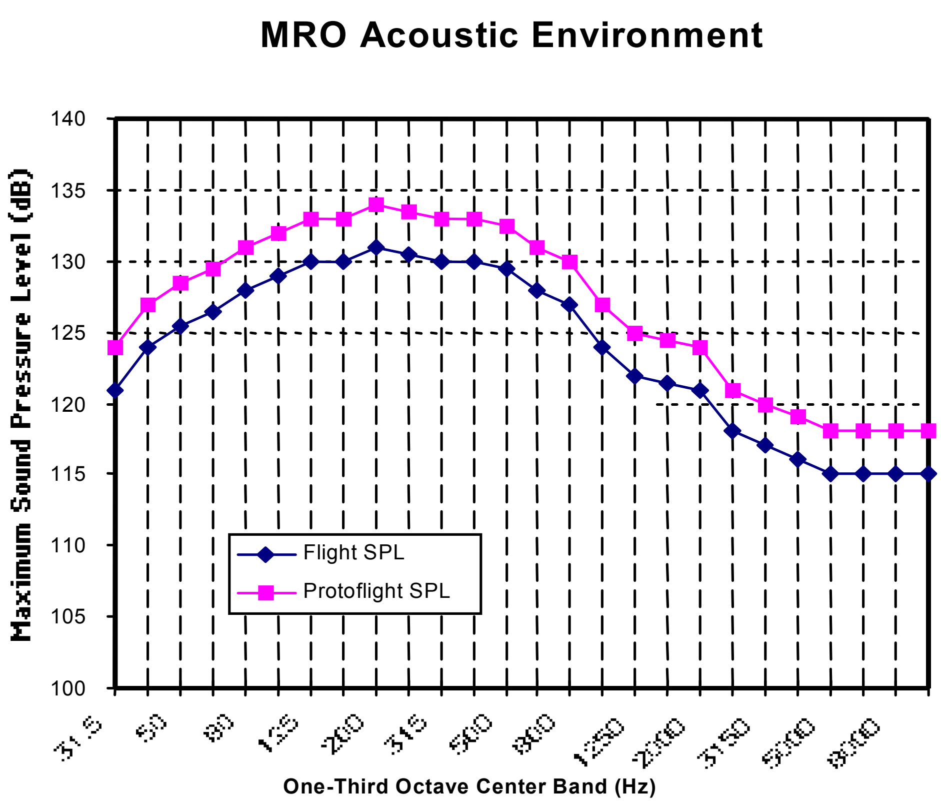


Figure 3-1 MRO Launch Acoustics Spectrum

Table 3-1 MRO Launch Acoustics Spectrum

1/3 Oct. Band Center Frequency, Hz	Flight SPL, dB (Ref. 2 x 10^{-5} Pascal)	Protoflight SPL, dB (Ref. 2 x 10^{-5} Pascal)
31.5	121	124
40	124	127
50	125.5	128.5
63	126.5	129.5
80	128	131
100	129	132
125	130	133
160	130	133
200	131	134
250	130.5	133.5
315	130	133
400	130	133
500	129.5	132.5
630	128	131
800	127	130
1000	124	127
1250	122	125
1600	121.5	124.5
2000	121	124
2500	118	121
3150	117	120
4000	116	119
5000	115	118
6300	115	118
8000	115	118
10000	115	118
Overall	141	144

3.1.2 Vibrations

3.1.2.1 Orbiter

Random and transient vibrations are transmitted to the Orbiter from the launch vehicle through the launch vehicle adapter. Although Delta and Atlas payload user manuals typically define this environment in terms of a swept sinusoidal vibration test, the preferred verification method for MRO is a random vibration test. A random vibration test shall be performed on the Orbiter for 60 seconds per axis in each of three orthogonal axes, one of which is the thrust axis. Preliminary, expected protoflight Orbiter Protoflight random vibration test levels are as in Table 3-2.

Table 3-2. Orbiter Random Vibration Protoflight Test Requirements

Frequency (Hz)	**Acceleration Spectral Density Level**
10 - 20 20 - 200	+ 3 dB / octave 0.03 g^2 / Hz
Overall	2.3 G rms

The random vibration test is expected to be force limited to reduce over-test at hard mounted resonance frequencies. The upper bound force spectrum in Table 3-3 may be used to limit the input acceleration to the Orbiter. Additional notching of the random vibration input levels at Orbiter resonances may be required during testing and should be based on a combination of the Mass Acceleration Curve (MAC) values of section 3.1.4 and the results of the Coupled Loads Analysis (CLA). These force and acceleration limit values may be modified based on information gathered during the shaker testing. The random vibration levels are not intended for use in the design of Orbiter primary structure or for the structural integrity of equipment supports.

Table 3-3. Orbiter Random Vibration Force Limit Specifications

Frequency (Hz)	Force Spectral Density Level
10 - 100 100 - 200	6 W N^2 / Hz - 6 dB / Octave
where W is the square of the Orbiter mass in kg.	

3.1.2.2 Assemblies

Assembly and Payload random vibration test criteria are derived from the Orbiter level acoustic and random vibration test requirements. Random vibration tests are required on all Orbiter assemblies, including Payloads, for one minute per axis in each of three orthogonal axes prior to delivery for Orbiter integration. Assembly and Payload interface random vibration test levels are currently TBD. For protoflight testing, 3.0 dB is added across the spectrum to the maximum predicted flight level, as stated in Table A-1. Planning levels for protoflight test are provided in Table 3-4.

Table 3-4. Assembly and Payload Planning Levels for Protoflight Random Vibration Testing

Frequency (Hz)	**Protoflight Test Level**
20 - 50 50 - 600 600 - 2000	+ 3 dB / octave 0.2 g^2/Hz - 3 dB / octave
Overall	16.1 G rms

For minimum workmanship purposes, assembly and Payload random vibration test requirements shall not be specified lower than given in Table 3-5.

Table 3-5. Assembly and Payload Minimum Workmanship Random Vibration Test Levels

Frequency (Hz)	**MinimumWorkmanship Level**
20 – 80 80 – 500 500- 1000	+3 dB/Oct 0.04 G^2/Hz -3 dB/Oct
Overall Level	6.8 Grms

Swept sinusoidal vibration tests are not required on assemblies and Payloads with fundamental frequencies above 80 Hz. Sinusoidal vibration criteria shall be specially derived for assemblies and Payloads with fundamental frequencies below 80 Hz.

Assembly and Payload random vibration tests may be force limited to reduce over-test at hard mounted resonance frequencies. The upper bound force spectrum below may be used to limit the input acceleration. The force limit values may be modified based on information gathered during shaker testing. In testing, responses of assemblies with a first resonance below 80 Hz in an axis may also be limited to the limit load value (see section 3.1.4) times 1.20. Preliminary, upper bound Assembly and Payload force limit values are provided in Table 3-6 below.

Table 3-6. Assembly Random Vibration Force Limit Specifications

Frequency, Hz	Force Spectral Density Level
20 – 1.1 Fn	480 F N^2 / Hz
1.1 Fn - 1000	- 6 dB / Oct.

The value, Fn, is the first predominant resonance frequency in the axis of test. The factor, F, is the product of the acceleration spectrum times the square of the total mass of the assembly in Kg.

3.1.3 Pyroshock

The Orbiter will experience pyroshocks due to firing of pyrotechnic devices during separation and deployment events. Shock levels at the spacecraft adapter interface due to other flight shock events such as launch vehicle stage separations, fairing jettison and engine ignition/shutdown are not significant compared to the spacecraft separation shock environment. The separation shock environments are defined in terms of shock response spectrum (SRS) for a frequency range of 100 to 10,000 Hz. The spacecraft shall meet its performance requirements after exposure to the maximum expected separation shocks as given in Table 3-7 or at least two actual firings of the separation band.

Assembly and Payload pyrotechnic shock levels are currently TBD. Pyroshock tests shall be performed on selected assemblies and Payloads. The shock pulses, with their shock response spectra corresponding to the specification, shall be applied at the assembly mounting points in each of three orthogonal axes. The synthesized shock waveform shall satisfy both of the following criteria: a) The pulse will be oscillatory in nature, b) The pulse will decay to less than 10% of its peak value within 20 milliseconds. For protoflight testing, 3.0 dB is added across the spectrum to the maximum predicted flight level, as stated in Table A-1.

Table 3-7. Orbiter Pyrotechnic Shock Environments

Frequency, Hz	Peak SRS Response (Q=10)
100	50 g
100 - 1,600	+ 10.0 dB / Oct.
1,600 - 10,000	5,000 g

1g = standard acceleration due to gravity = 9.81 m/s^2

3.1.4 Quasi-static Loads

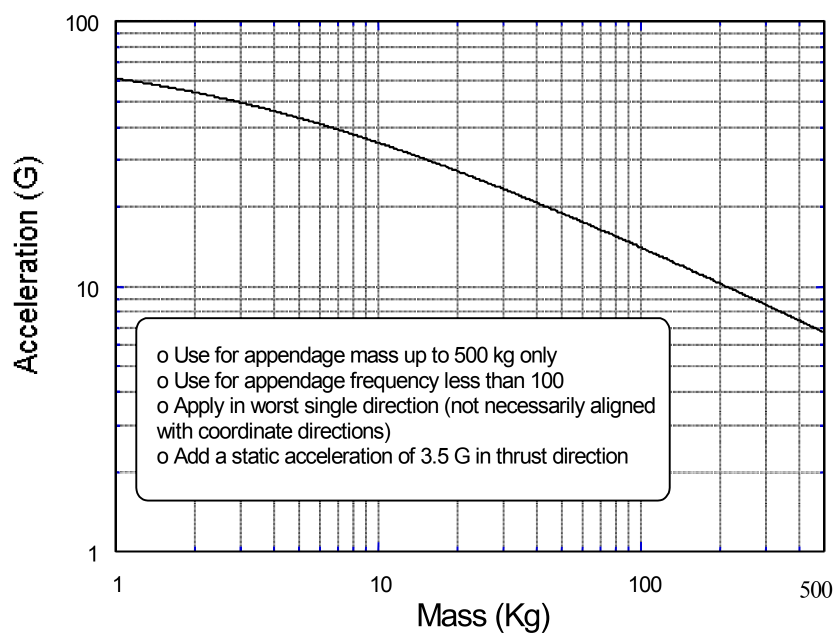
Quasi-static loads specified for structural design and qualification account for the combined effects of quasi-steady accelerations due to the launch vehicle thrust and of low frequency vibrations due to launch vehicle transient and random excitation events. For preliminary design the limit load factors given in Table 3-1 may be used to evaluate the spacecraft primary structure, and all other structure shall be designed to the limit load values derived from the Mass Acceleration Curve (MAC) given in Figure 3-2. Limit load values shall be updated for primary structure from the Orbiter / launch vehicle coupled loads analysis when available. The preliminary design limit load accelerations presented in Table 3-8 are applicable at the Orbiter's center of gravity (C.G). The MAC shall be the limit load criteria for secondary structures and the structural support of assemblies and Payloads and is applied in the most adverse direction, then superimposed with a static 3.5 g in the thrust direction. For design, the limit load value shall be multiplied by a factor of safety of 1.25 for yield and 1.4 for ultimate. Primary structure shall be tested to 1.2 times its limit load.

Table 3-8. Orbiter Preliminary Design Limit-Load Accelerations (Unit: g) (1)

Load Condition (2)	Lift-off/Transonic/1st Stage Burnout			
	Case 1	Case 2	Case 3	Case 4
Lateral Axes	± 0.5	± 3.0	± 3.0	± 0.5
Thrust Axis (3)	+6.5	+3.5	-1.0	-1.0

(1) Limit load accelerations should be multiplied by appropriate safety factors to obtain structural design loads.
(2) Lateral and longitudinal loading may act simultaneously during any flight event.
(3) Plus indicates compression loads and minus indicates tension load.

Figure 3-2. Orbiter Preliminary Physical Mass Acceleration Curve (MAC)



3.1.5 Explosive Atmosphere

The Orbiter shall be designed to operate in the presence of flammable vapors without initiating an explosion or fire, in accordance with JPL D-560, JPL Standard for Systems Safety, paragraph 2.3.2.

SECTION 4
CRUISE ENVIRONMENT

The flight environmental estimates specified in this section comprise those environments that the hardware may encounter during cruise.

4.1 THERMAL RADIATION

The minimum expected thermal radiation level is 3.0 x 10^{-6} W/m^2 at a source effective temperature of 2.7 K. Expected thermal radiation levels are given in Table 4-1

The percentage of the solar constant associated with wavelengths in the range of 0.85 to 7.0 micrometers is given in Table 4-2. The percentages are expected to remain unchanged for planetary reflected solar radiation. The relative spectral distribution for planetary infrared (IR) is represented by a Plank distribution derived using the stated effective blackbody temperatures

Solar radiation varies inversely with the square of the distance from the sun. The solar flux at 1.0 astronomical unit is 1367 W/m^2 $\pm$ 1.5%.

Table 4-1 Thermal Radiation Environments

Mission Phase	**Direct Solar**	**Reflected Solar (albedo)**	**Planetary IR (LW Radiation)**
Earth Orbit:	0 to 1400 W/m^2 (5770K effective blackbody temperature)	0 to 0.32 0 to 450 W/m^2 (global annual mean) 0 to 0.70 (polar regions)	0 to 270 W/m^2 (206K to 262K effective blackbody temperature)
Earth/Mars Cruise: Near Earth	0 to 1414 W/m^2 (at earth perihelion)	Negligible beyond 4 earth radii	Negligible beyond 4 earth radii
Mars Perihelion	0 to 710 W/m^2	Negligible beyond 4 Mars radii	Negligible beyond 4 Mars radii
Mars Aphelion	0 to 490 W/m^2	Negligible beyond 4 Mars radii	Negligible beyond 4 Mars radii
Mars Orbit: Mars Perihelion	0 to 710 W/m^2	See Table 5-1	128 W/m^2
Mars Aphelion	0 to 490 W/m^2	See Table 5-1	99 W/m^2
Mars Occultation	0	0	as above

Note: Due to Orbiter orientation, a surface may not see direct solar, reflected solar or planetary infrared. The percentage of the solar constant associated with wavelengths in the range of 0.085 to 7.0 micrometers is given in Table 4-2. The percentages are expected to remain unchanged for planetary reflected solar radiation. The relative spectral distribution for planetary infrared (IR) is represented by a Planck distribution consistent with the provided planetary IR fluxes.

Table 4-2. Solar-Spectral-Irradiance Data, 0.0850 to 7.0 Micrometers

λ (μm)	P (%)	λ (μm)	P (%)	λ (μm)	P (%)
0.0850	3.8×10^{-4}	0.36	5.317	0.67	43.745
0.0900	3.9×10^{-4}	0.365	5.723	0.68	44.816
0.0950	4.0×10^{-4}	0.37	6.151	0.69	45.856
0.1000	4.1×10^{-4}	0.375	6.583	0.70	46.880
0.1050	4.2×10^{-4}	0.38	7.003	0.71	47.882
0.1100	4.2×10^{-4}	0.385	7.413	0.72	48.865
0.1150	4.3×10^{-4}	0.39	7.819	0.73	49.827
0.1200	4.4×10^{-4}	0.395	8.242	0.74	50.769
0.1250	4.7×10^{-4}	0.40	8.725	0.75	51.691
0.1320	4.9×10^{-4}	0.405	9.293	0.80	56.019
0.1350	5.2×10^{-4}	0.41	9.920	0.85	59.890
0.1400	5.4×10^{-4}	0.415	10.572	0.90	63.358
0.1450	5.6×10^{-4}	0.42	11.222	0.95	66.544
0.1500	5.8×10^{-4}	0.425	11.858	1.0	69.465
0.1550	6.3×10^{-4}	0.43	12.474	1.1	74.409
0.1600	6.9×10^{-4}	0.435	13.084	1.2	78.386
0.1650	8.2×10^{-4}	0.44	13.726	1.3	81.638
0.1700	1.01×10^{-3}	0.445	14.415	1.4	84.343
0.1750	1.31×10^{-3}	0.45	15.141	1.5	86.645
0.1800	1.70×10^{-3}	0.455	15.892	1.6	88.607
0.1850	2.33×10^{-3}	0.46	16.653	1.7	90.256
0.1900	3.16×10^{-3}	0.465	17.414	1.8	91.590
0.1950	5.2×10^{-3}	0.47	18.168	1.9	92.643
0.2000	8.1×10^{-3}	0.475	18.921	2.0	93.489
0.2050	1.34×10^{-2}	0.48	19.682	2.1	94.202
0.2100	2.05×10^{-2}	0.485	20.430	2.2	94.827
0.2150	3.53×10^{-2}	0.49	21.156	2.3	95.370
0.22	0.0502	0.495	21.878	2.4	95.858
0.225	0.0729	0.50	22.599	2.5	96.294
0.23	0.0972	0.505	23.313	2.6	96.671
0.235	0.1205	0.51	24.015	2.7	97.007
0.24	0.1430	0.515	24.702	2.8	97.310
0.245	0.1681	0.52	25.379	2.9	97.584
0.25	0.1944	0.525	26.060	3.0	97.828
0.255	0.2267	0.53	26.743	3.1	98.038
0.26	0.270	0.535	29.419	3.2	98.218
0.265	0.328	0.54	28.084	3.3	98.372
0.27	0.405	0.545	28.738	3.4	98.505
0.275	0.486	0.55	29.381	3.5	98.620
0.28	0.465	0.555	30.017	3.6	98.725
0.285	0.644	0.56	30.648	3.7	98.819
0.29	0.811	0.565	31.276	3.8	98.906
0.295	1.008	0.57	31.908	3.9	98.985
0.30	1.211	0.575	32.542	4.0	99.058
0.305	1.417	0.58	33.176	4.1	99.125
0.31	1.656	0.585	33.809	4.2	99.186
0.315	1.924	0.59	34.440	4.3	99.241
0.32	2.219	0.595	35.065	4.4	99.291
0.325	2.552	0.60	35.683	4.5	99.337
0.33	2.928	0.61	36.902	4.6	99.379
0.335	3.324	0.62	38.098	4.7	99.416
0.34	3.722	0.63	39.270	4.8	99.450
0.345	4.118	0.64	40.421	4.9	99.482
0.35	4.517	0.65	41.550	5.0	99.511
0.355	4.919	0.66	42.658	6.0	99.718
				7.0	99.819

λ (μm) is wavelength; and P is the percentage of the solar constant associated with wavelengths shorter than λ.

4.2 ASSEMBLY TEMPERATURES

Orbiter temperature interface predictions and Allowable Flight Temperatures (AFT) for assemblies (including Payloads) shall be derived from the Orbiter thermal design. Protoflight test temperatures shall be defined for all assemblies and Payloads based on the requirements stated in Table A-1, Appendix A. Appendix B contains definitions of Allowable Flight Temperature (AFT), Protoflight Test (PF), Flight Acceptance Test (FA), etc.

4.3 ELECTROMAGNETIC INTERFERENCE

Requirements are covered by the ground operations requirements in Section 2.3

4.4 HIGH ENERGY RADIATION ENVIRONMENTS

Unless otherwise stated, all tables and graphs within this section represent environments external to the Orbiter, and do not contain a design factor. The Radiation Design Factor (RDF) is defined as

$$\text{RDF} = \frac{\text{Radiation-resisting capability of a part or component in a given application}}{\text{Radiation environment present at the location of the part or component}}$$

4.4.1 Total Ionizing Dose (TID)

The total ionizing dose (TID) exposure of MRO mission hardware will come from solar protons, from solar and galactic cosmic rays. The TID contribution from solar protons is expected to dominate for all hardware.

A subsystem's electronic devices shall be chosen such that the subsystem operates within performance specification during and after the exposure of solar protons (Table 4-3 and Figure 4-1) at a radiation design factor (RDF) of 2 times the TID level present at the location of the device. A device's solar proton TID exposure is provided in Table 4-4 and Figure 4-2, which give the solar proton TID at the center of an aluminum spherical shield geometry.

Devices that require spot shielding shall be assessed at a radiation design factor (RDF) of 3 times the TID level present at the location of the device.

4.4.1.1. Solar Protons

Table 4-3 and Figure 4-1 show the expected mission fluence, and Table 4-4 & Figure 4-2 show the expected mission TID of solar protons, derived from a statistical model [Feynman, et al. 1991 model] based on data from approximately 3 solar cycles. A 95th percentile confidence level is used. Fluences are based on the following approximate mission schedule: Launch: August 2005; Arrival at Mars: March 2006; Aerobraking to mapping orbit (400km): through October 2006; Science Mission: December 2006 to November 2008; Possible Extended Science Mission: December 2008 to November 2009; Relay Mission: December 2009 to December 2010.

4.4.1.2 Electrons

The Flux of Interplanetary Energetic Electrons is usually an order of magnitude lower than protons and therefore do not contribute a significant radiation dose. For the MRO mission, energetic electrons can be neglected.

4.4.1.3 Galactic Cosmic Rays

The flux of galactic cosmic rays (GCR) contributes approximately 0.02 Rad-Si /day to the mission dose, independent of shielding. For the MRO mission, GCRs will be neglected.

4.4.2 Displacement Damage

The radiation degradation of certain electronic devices (solar cells and opto-couplers, among others), cannot be adequately characterized in terms of TID. The particle type, energy, and fluence must be considered.

Subsystems' electronic devices shall be chosen with consideration of displacement damage accumulation such that the subsystem operates within performance specification during and after exposure to two times what remains of the solar proton fluence specified in Table 4-3, after having passed through the shielding around the device.

4.4.3. Single Event Effects

Electronics may be susceptible to Single Event Effects, or SEE, which include reversible, non-destructive actions (Single Event Upsets, or SEUs) such as memory bit-flips; or potentially destructive actions such as device latch-up. SEEs are caused by high-energy ions. The term "heavy ion", as used below, refers to any ion having atomic number $Z > 1$; i.e. anything larger than a proton. If the part's SEE threshold LET (linear energy transfer) is less than 15 MeV-cm^2/mg, then high-energy protons can also cause SEE. These types of high-energy particles are found in galactic cosmic rays and solar particle events.

In electronic sensors, SEE can manifest itself as spurious signals, i.e. radiation-induced background noise.

A subsystem's electronic devices shall be chosen such that the subsystem operates within performance specification during and after exposure to the high-energy radiation environments specified in the following subsections. The subsystem and system-level requirements regarding performance with respect to SEE during operation are as follows:

1) Temporary loss of function or loss of data shall be permitted provided that the loss does not compromise subsystem/system health, full performance can be recovered rapidly, and there is no time in the mission that the loss is mission critical.

2) Normal operation and function shall be restored via internal correction methods without external intervention in the event of a SEU.

3) Fault traceability shall be provided in the telemetry stream to the greatest extent practical for all anomalies involving SEEs.

An RDF = 1 shall be applied to the environments specified in the following sub-sections.

4.4.3.1. Solar Proton Peak Flux

The solar proton peak flux environment, which is to be used for proton-induced SEE in parts having an SEE threshold less than 15 MeV-cm^2/mg, is given by the CREME96 model for the worst-case (5 minute average) solar event protons as seen behind 25 mils of aluminum shielding. This is shown in Figure 4-3.

4.4.3.2. Solar Heavy Ion Peak Flux

The solar particle event heavy ion peak flux environment is given by the CREME96 model for the worst case (5 minute average) heavy ions as seen behind 25 mils of aluminum shielding. This is shown in Figure 4-4.

4.4.3.3. Galactic Cosmic Ray Proton Flux

The GCR proton peak flux environment, to be used for proton-induced SEE in parts having an SEE threshold less than 15 MeV-cm^2/mg, is given by the CREME96 model for GCR protons at solar minimum as seen behind 25 mils aluminum shielding. This is shown in Figure 4-3.

4.4.3.4. Galactic Cosmic Ray Heavy Ion Flux

The GCR heavy ion peak flux environment is given by the CREME96 model for heavy ions at solar minimum as seen behind 25 mils aluminum shielding. This is shown in Figure 4-4.

Table 4-3 - Solar Flare Proton Fluence at 95% Confidence Level

(Cumulative through Cruise + Aerobraking, through Science, through Extended Science, and through Relay Mission: 5 years overall)

Energy (MeV)	FLUENCE (Protons/cm^2)			
	Cruise + Aerobrake (ending approx Nov 2006)	Science (ending approx Nov 2008)	Extended Science (ending approx Nov 2009)	Relay (ending approx Nov 2010)
1	9.35E+10	9.35E+10	1.62E+11	2.44E+11
4	3.13E+10	3.13E+10	5.43E+10	8.18E+10
10	1.21E+10	1.21E+10	2.10E+10	3.16E+10
30	3.31E+09	3.31E+09	5.74E+09	8.66E+09
60	1.49E+09	1.49E+09	2.58E+09	3.89E+09
100	8.27E+08	8.27E+08	1.43E+09	2.16E+09
150	5.17E+08	5.17E+08	8.96E+08	1.35E+09
200	3.71E+08	3.71E+08	6.42E+08	9.68E+08
300	2.32E+08	2.32E+08	4.02E+08	6.06E+08
500	1.29E+08	1.29E+08	2.23E+08	3.37E+08
1000	5.76E+07	5.76E+07	9.98E+07	1.50E+08
2000	2.60E+07	2.60E+07	4.50E+07	6.78E+07
3000	5.70E-01	5.70E-01	9.88E-01	1.49E+00

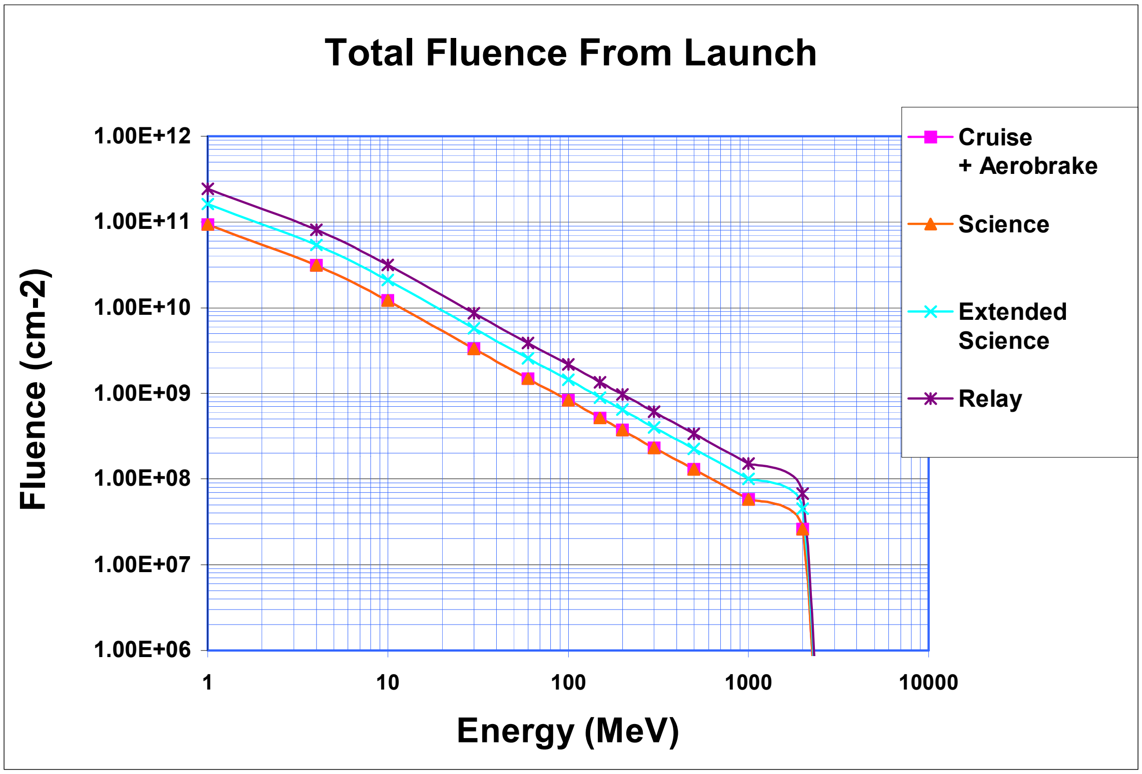


Figure 4-1 Total Solar Flare Proton Fluence at 95% Confidence Level

Table 4-4 MRO Total Ionizing Dose, Cumulative to End of Each Mission Segment
(Table includes RDF = 2)

Aluminum Shielding Thickness			Cruise + Aerobrake	Science	Extended Science	Relay
mm	gcm^{-2}	mils	Rads(Si) Total Ionizing Dose Includes RDF=2			
3.70E-02	1.00E-02	1.46E+00	1.83E+05	1.83E+05	3.16E+05	4.77E+05
4.66E-02	1.26E-02	1.84E+00	1.48E+05	1.48E+05	2.57E+05	3.87E+05
5.87E-02	1.59E-02	2.31E+00	1.21E+05	1.21E+05	2.10E+05	3.16E+05
7.39E-02	2.00E-02	2.91E+00	1.01E+05	1.01E+05	1.76E+05	2.65E+05
9.30E-02	2.51E-02	3.66E+00	8.36E+04	8.36E+04	1.45E+05	2.18E+05
1.17E-01	3.16E-02	4.61E+00	7.11E+04	7.11E+04	1.23E+05	1.86E+05
1.47E-01	3.98E-02	5.81E+00	5.94E+04	5.94E+04	1.03E+05	1.55E+05
1.86E-01	5.01E-02	7.31E+00	4.72E+04	4.72E+04	8.18E+04	1.23E+05
2.34E-01	6.31E-02	9.20E+00	3.83E+04	3.83E+04	6.64E+04	1.00E+05
2.54E-01	6.86E-02	1.00E+01	3.39E+04	3.39E+04	5.87E+04	8.85E+04
2.94E-01	7.94E-02	1.16E+01	2.91E+04	2.91E+04	5.04E+04	7.60E+04
3.70E-01	1.00E-01	1.46E+01	2.36E+04	2.36E+04	4.09E+04	6.17E+04
4.66E-01	1.26E-01	1.84E+01	1.92E+04	1.92E+04	3.32E+04	5.01E+04
5.08E-01	1.37E-01	2.00E+01	1.72E+04	1.72E+04	2.99E+04	4.50E+04
5.87E-01	1.59E-01	2.31E+01	1.51E+04	1.51E+04	2.61E+04	3.93E+04
7.39E-01	2.00E-01	2.91E+01	1.23E+04	1.23E+04	2.14E+04	3.22E+04
7.62E-01	2.06E-01	3.00E+01	1.14E+04	1.14E+04	1.97E+04	2.97E+04
9.30E-01	2.51E-01	3.66E+01	9.25E+03	9.25E+03	1.60E+04	2.42E+04
1.02E+00	2.74E-01	4.00E+01	8.35E+03	8.35E+03	1.45E+04	2.18E+04
1.17E+00	3.16E-01	4.61E+01	7.41E+03	7.41E+03	1.28E+04	1.94E+04
1.27E+00	3.43E-01	5.00E+01	6.56E+03	6.56E+03	1.14E+04	1.71E+04
1.47E+00	3.98E-01	5.81E+01	5.70E+03	5.70E+03	9.88E+03	1.49E+04
1.52E+00	4.12E-01	6.00E+01	5.27E+03	5.27E+03	9.13E+03	1.38E+04
1.78E+00	4.80E-01	7.00E+01	4.69E+03	4.69E+03	8.13E+03	1.22E+04
1.86E+00	5.01E-01	7.31E+01	4.32E+03	4.32E+03	7.49E+03	1.13E+04
2.03E+00	5.49E-01	8.00E+01	3.91E+03	3.91E+03	6.78E+03	1.02E+04
2.29E+00	6.17E-01	9.00E+01	3.55E+03	3.55E+03	6.15E+03	9.27E+03
2.34E+00	6.31E-01	9.20E+01	3.33E+03	3.33E+03	5.77E+03	8.70E+03
2.54E+00	6.86E-01	1.00E+02	3.06E+03	3.06E+03	5.30E+03	7.98E+03
2.94E+00	7.94E-01	1.16E+02	2.69E+03	2.69E+03	4.67E+03	7.03E+03
3.05E+00	8.23E-01	1.20E+02	2.46E+03	2.46E+03	4.27E+03	6.44E+03
3.56E+00	9.60E-01	1.40E+02	2.18E+03	2.18E+03	3.78E+03	5.69E+03
3.70E+00	1.00E+00	1.46E+02	2.02E+03	2.02E+03	3.50E+03	5.27E+03
4.06E+00	1.10E+00	1.60E+02	1.85E+03	1.85E+03	3.20E+03	4.83E+03
4.57E+00	1.23E+00	1.80E+02	1.65E+03	1.65E+03	2.87E+03	4.32E+03
4.66E+00	1.26E+00	1.84E+02	1.54E+03	1.54E+03	2.67E+03	4.02E+03
5.08E+00	1.37E+00	2.00E+02	1.41E+03	1.41E+03	2.45E+03	3.69E+03
5.59E+00	1.51E+00	2.20E+02	1.32E+03	1.32E+03	2.29E+03	3.46E+03
5.87E+00	1.59E+00	2.31E+02	1.25E+03	1.25E+03	2.17E+03	3.28E+03
6.10E+00	1.65E+00	2.40E+02	1.17E+03	1.17E+03	2.04E+03	3.07E+03
6.60E+00	1.78E+00	2.60E+02	1.09E+03	1.09E+03	1.89E+03	2.85E+03
7.11E+00	1.92E+00	2.80E+02	9.91E+02	9.91E+02	1.72E+03	2.59E+03
7.39E+00	2.00E+00	2.91E+02	9.39E+02	9.39E+02	1.63E+03	2.45E+03
7.62E+00	2.06E+00	3.00E+02	8.90E+02	8.90E+02	1.54E+03	2.32E+03
9.30E+00	2.51E+00	3.66E+02	7.71E+02	7.71E+02	1.34E+03	2.01E+03

Table 4-4 MRO Total Ionizing Dose, Cumulative to End of Each Mission Segment
(Table includes RDF = 2)
-- continued --

Aluminum Shielding Thickness			Cruise + Aerobrake	Science	Extended Science	Relay
mm	gcm^{-2}	mils	Rads(Si) Total Ionizing Dose Includes RDF=2			
1.02E+01	2.74E+00	4.00E+02	6.80E+02	6.80E+02	1.18E+03	1.78E+03
1.17E+01	3.16E+00	4.61E+02	5.91E+02	5.91E+02	1.02E+03	1.54E+03
1.27E+01	3.43E+00	5.00E+02	5.29E+02	5.29E+02	9.17E+02	1.38E+03
1.47E+01	3.98E+00	5.81E+02	4.57E+02	4.57E+02	7.93E+02	1.19E+03
1.86E+01	5.01E+00	7.31E+02	3.63E+02	3.63E+02	6.29E+02	9.47E+02
2.34E+01	6.31E+00	9.20E+02	2.83E+02	2.83E+02	4.90E+02	7.39E+02
2.94E+01	7.94E+00	1.16E+03	2.24E+02	2.24E+02	3.87E+02	5.84E+02
3.70E+01	1.00E+01	1.46E+03	1.70E+02	1.70E+02	2.95E+02	4.44E+02

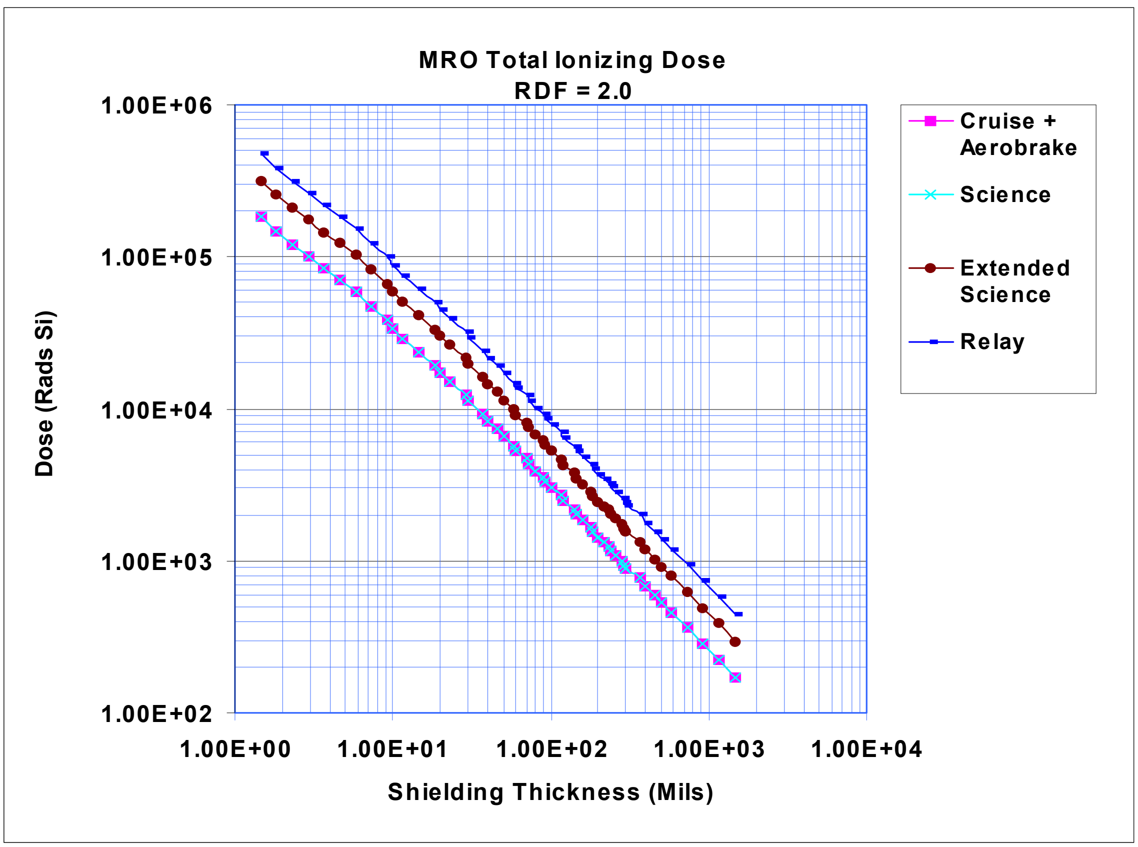


Figure 4-2 Total Ionizing dose for MRO Mission
(RDF = 2 is included)

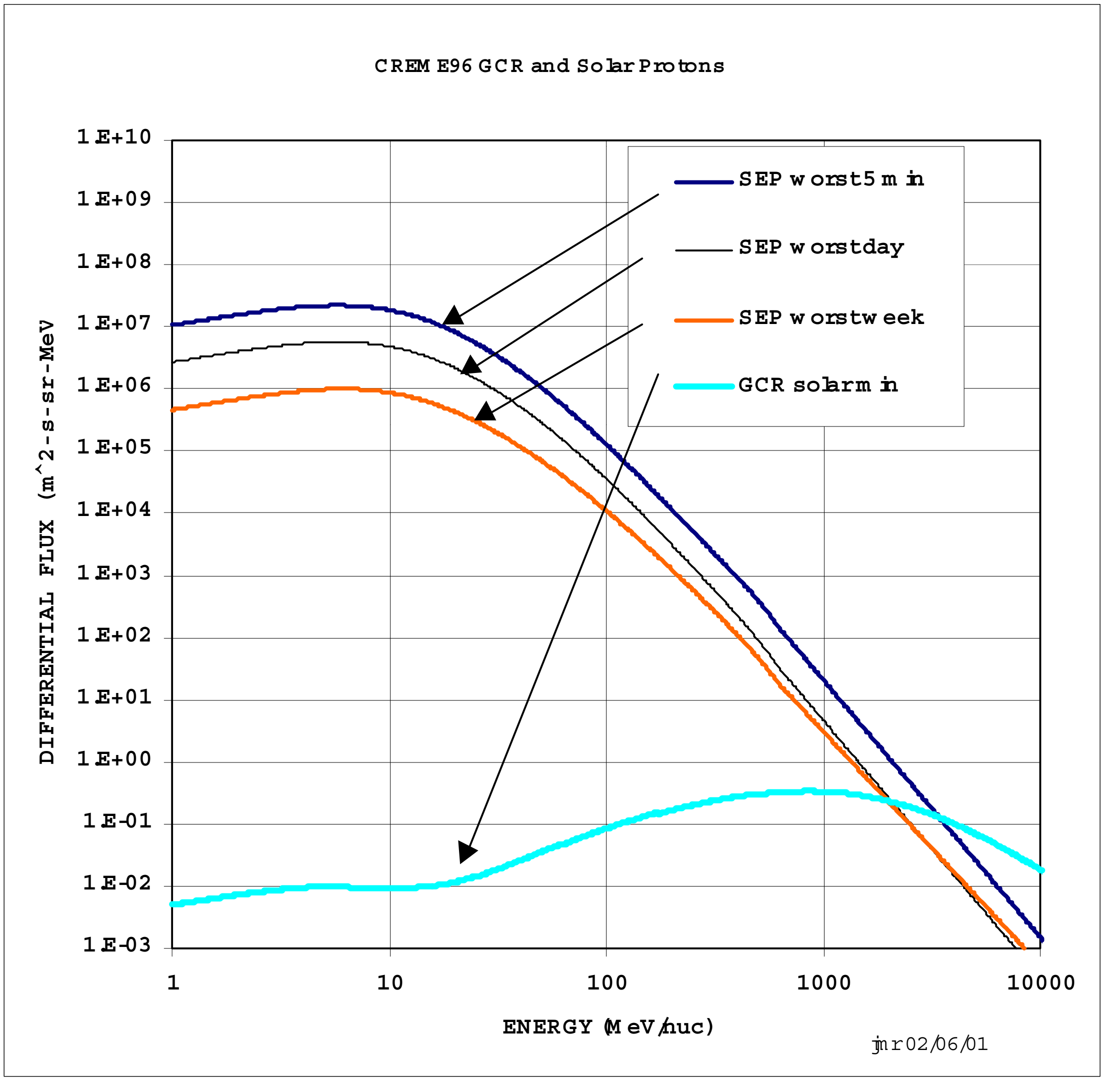


Figure 4-3 Solar Energetic Particle (SEP) event and GCR proton fluxes behind 25 mils aluminum shielding (CREME96 model), to be used for determining SEE rates.

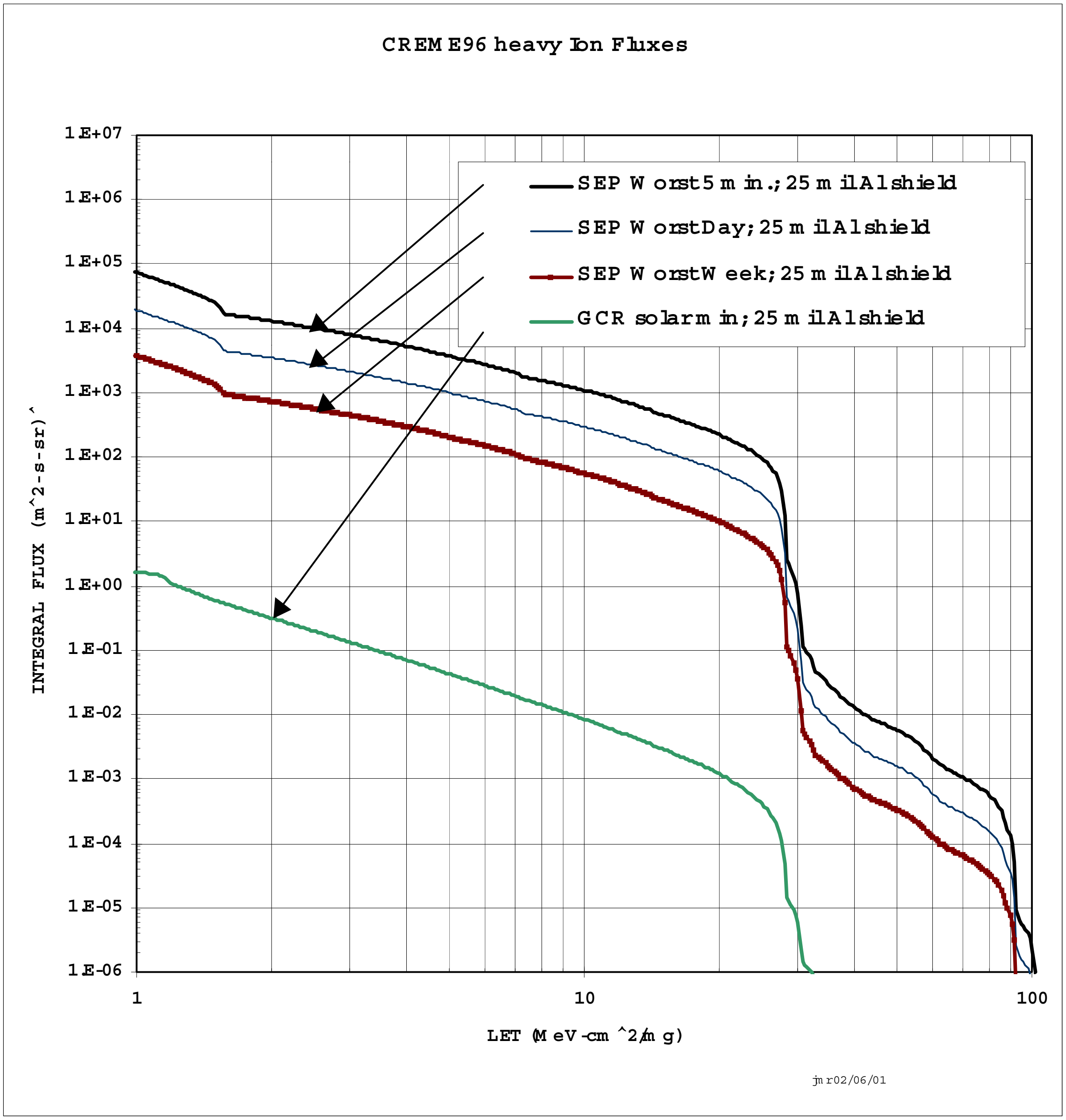


Figure 4-4 Solar Energetic Particle (SEP) event and GCR heavy ion fluxes behind 25 mils aluminum shielding (CREME96 model), to be used for determining SEE rates.

4.5 MICROMETEOROID OMNIDIRECTIONAL FLUENCE

The micrometeoroid environment is provided in Table 4-5 which shows the omnidirectional fluence - the number of particles per unit area that impact the surface of a spherical object, averaged over all directions, and assuming normal incidence impact angle. (When divided by 4, the omnidirectional fluence gives the fluence to one side of a surface, averaged over all orientations of the surface.)

The Orbiter, including Payloads, designs shall be such that the probability of no critical damage from micrometeoroid impact shall be >0.95. Particular attention should be paid to pressure vessels and lines, cables, solar cells, and optics.

Table 4-5 Micrometeoroid Environment

Mass (grams)	Mission Omnidirectional Fluence (m^{-2})
1.E-12	1.34E+04
1.E-11	6.27E+04
1.E-10	2.84E+03
1.E-09	1.29E+03
1.E-08	5.13E+02
1.E-07	1.24E+02
1.E-06	2.04E+01
1.E-05	3.12E+00
1.E-04	4.71E-01
1.E-03	4.00E-02
1.E-02	2.26E-03
1.E-01	1.07E-04
1.E+00	5.00E-06
1.E+01	2.33E-07

Mean penetration velocity = 20 Km/sec
Particle mass density = 2.5 grams/cm^3

4.6 MAGNETIC FIELD

Magnetic field strengths are expected to be in the range from $5x10^4$ nT (0.5 gauss) near earth to 25 nT (0.00025 gauss) during cruise.

4.7 DYNAMICS

Cruise dynamic environments may consist of pyroshock from actuation of pyro devices, quasi-static acceleration from thruster firings and low-level vibration from thruster firings and on-board equipment such as reaction wheels, motors or actuators, etc. These environments shall be estimated based on the Orbiter design.

SECTION 5
MARS AEROBRAKING AND ON-ORBIT ENVIRONMENTS

5.1 THERMAL RADIATION.

Direct solar fluxes in Mars Orbit are shown in Table 4-1. The percentage of the solar flux associated with wavelengths in the range of 0.0850 to 7.0 micrometers is given in Table 4-2. The percentages are expected to remain unchanged for planetary reflected solar radiation. The relative spectral distribution for planetary infrared (IR) is represented by a Plank distribution using the stated effective blackbody temperatures. Expected orbital variation of the planetary infrared fluxes for Mars polar orbit at perihelion and aphelion may be determined from the Mars surface temperatures given in Figures 5-1 and 5-2. . The emittance of the surface of Mars is taken as 1.0. The worst-case orbital variation of albedo fluxes may be determined from the albedos of Table 5-1.

Table 5-1. Mars Albedo Distribution

LATITUDE (deg)	PERIHELION ALBEDO (maximum albedo)	APHELION ALBEDO (minimum albedo)
80 to 90	0.5	0.3
70 to 80	0.5	0.2
60 to 70	0.5	0.2
50 to 60	0.5	0.17
40 to 50	0.28	0.17
30 to 40	0.28	0.18
20 to 30	0.28	0.22
10 to 20	0.28	0.25
0 to 10	0.28	0.25
-10 to 0	0.28	0.20
-20 to -10	0.25	0.18
-30 to -20	0.22	0.18
-40 to -30	0.22	0.18
-50 to -40	0.25	0.3
-60 to -50	0.25	0.4
-70 to -60	0.3	0.4
-80 to -70	0.4	0.4
-90 to -80	0.4	0.4

Solar declination at perihelion = -23.7 deg
Solar declination at aphelion = 23.7 deg

5.2 MAGNETIC FIELD.

The intrinsic Mars magnetic field is less than 2×10^3 nT (0.002 gauss).

5.3 MICROMETEOROID ENVIRONMENT.

The meteoroid environment near Mars is included in the fluence spectrum provided in the cruise phase estimate given in Section 4.5.

5.4 HIGH ENERGY CHARGED PARTICLE RADIATION.

The charged particle radiation environment at Mars is included in the interplanetary charged particle environment given in Section 4.4.

5.5 DYNAMICS ENVIRONMENT.

Mars orbit dynamic environments may consist of pyroshock from actuation of pyro devices, quasi-static acceleration from thruster firings and low-level vibration from thruster firings and on-board equipment such as reaction wheels, motors or actuators, etc. These environments, at the payload interfaces and other potentially susceptible hardware, shall be estimated based on the Orbiter design.

5.6 ELECTROMAGNETIC COMPATIBILITY.

Estimates of the electromagnetic and magnetic interference levels during the orbital phase of the mission are expected to be enveloped by those given for the ground operations and handling phase of the mission (Section 2.3).

5.7 AEROBRAKING ENVIRONMENT

This section describes the aerodynamic heating rates and forces encountered by the Orbiter during the atmospheric passes in an aerobraking mission. Aerobraking uses the drag force on the Orbiter to remove orbital energy and reduce propellant requirements. The aerobraking phase will include three subphases: walk-in, main phase (or steady state), and walk-out. The severity of the aerobraking environment will depend on the amount of orbital energy removed, the duration of the steady state subphase, and the Orbiter ballistic coefficient. The orbital energy removed is determined by the initial orbit period and the final period for aerobraking.

Figure 5-1 represents the transient heating rate profile for a representative aerobraking drag pass. To convert freestream heating to surface heating use the following expression:

$$Q_{heatsurf} = Q_{heatfreestream} * Ch * AC * Cos(Ainc)$$

where: $Q_{heatfreestream} = 0.5 * Rho * V^3$

Rho = instantaneous atmospheric density

V = instantaneous velocity with respect to the atmosphere

Ch = is the transition flowfield heat coefficient (value = 0.9),

AC = is the surface thermal accommodation coefficient (value = 0.95),

Ainc = Incidence Angle (measured from surface normal)

A typical aerobraking drag pass will have a duration that ranges from 300 seconds (in long period orbits) to 1200 seconds (in short period orbits). The drag duration is defined as the length of time the dynamic pressure is greater than 0.0015 N/m^2. The dynamic pressure (q) is defined as:

$$q = 0.5 * Rho * V^2$$

Rho = instantaneous atmospheric density
V = instantaneous velocity with respect to the atmosphere

Transient oscillations of ±10° in the orbital plane (pitch) and ±10° normal to the orbital plane (yaw) may occur early in each aerobraking pass.

For additional information regarding aerobraking, see the "MRO Mission and Trajectory Description Document."

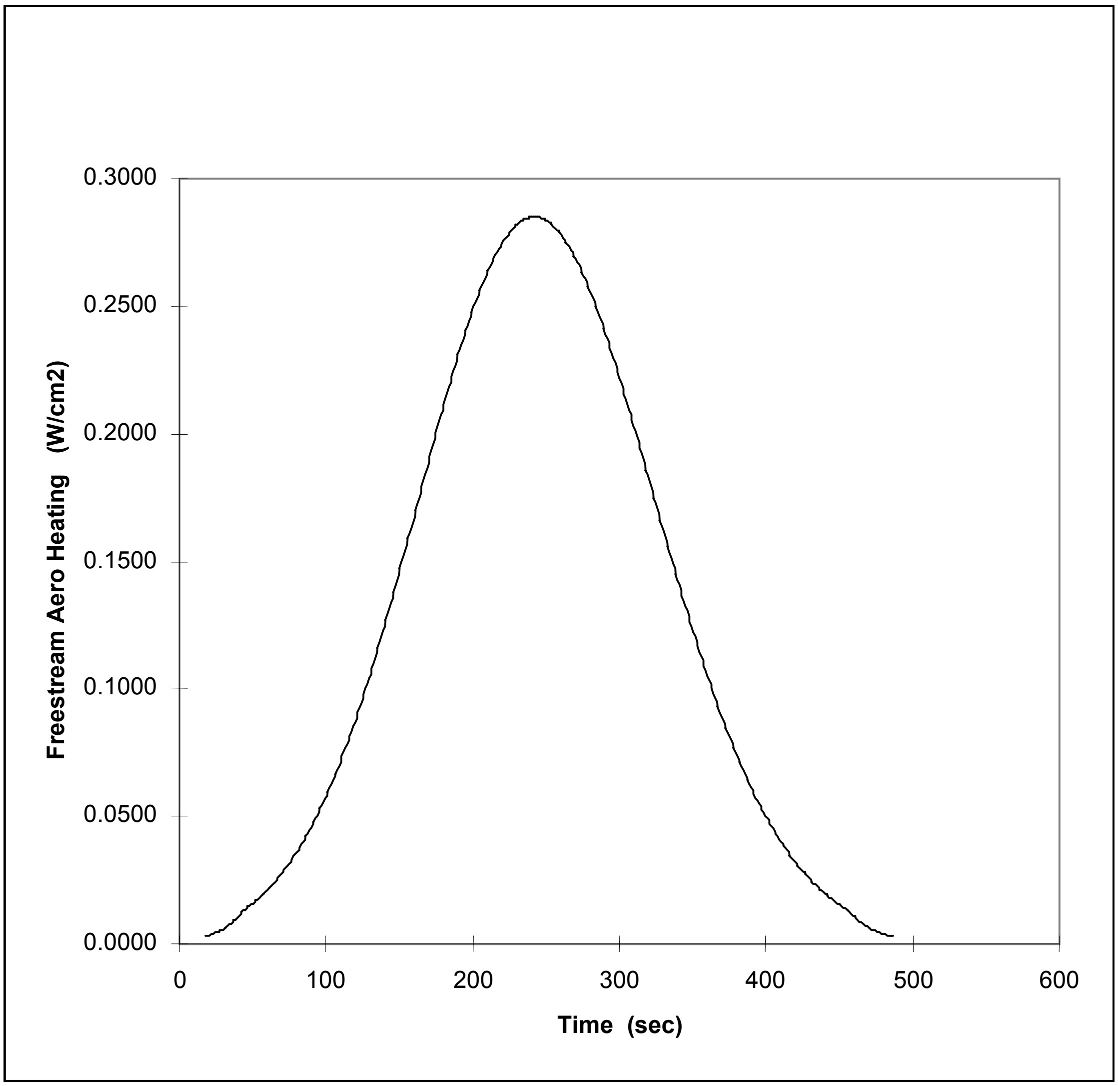


Figure 5-1 **Representative Heating Pulse**

5.8 ATOMIC OXYGEN

MRO surfaces shall be designed to tolerate an atomic oxygen (AO) fluence of $2x10^{21}$ atoms/cm^2.

APPENDIX A

ORBITER AND PAYLOAD ENVIRONMENTAL TEST MARGINS

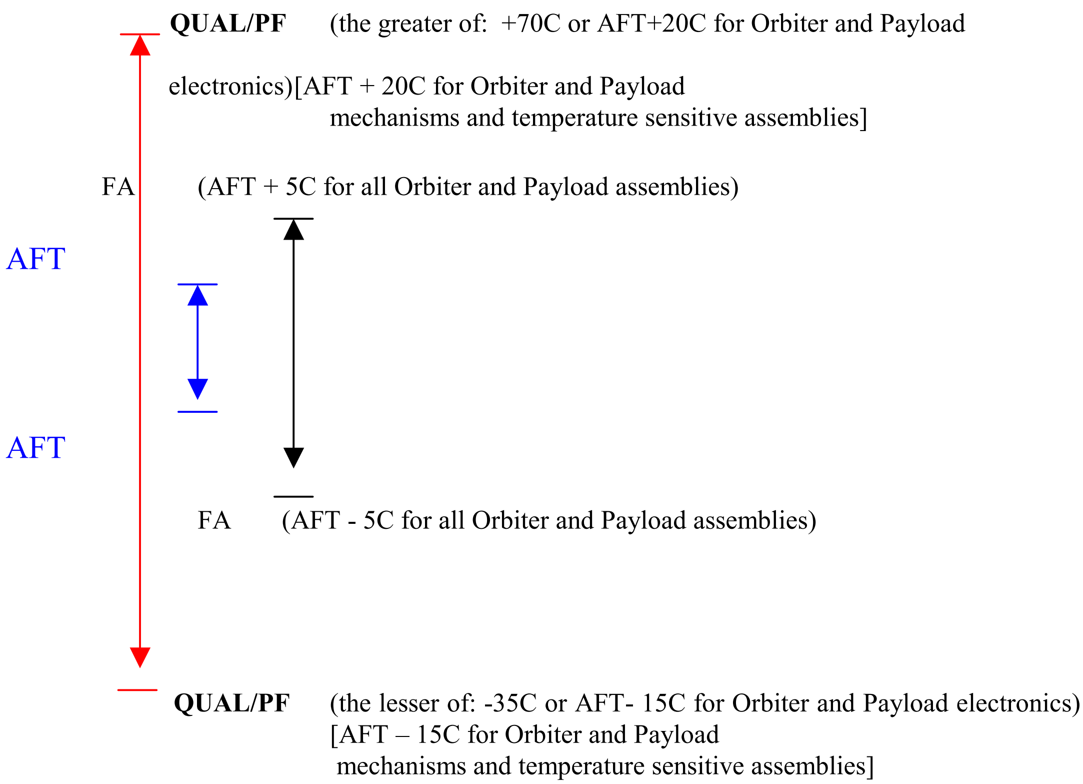
	Orbiter System	**Assembly Level and Payload**		
	Protoflight	**Design/-Qualification**	**Protoflight**	**Flight Acceptance**
Acoustics (1) Level Duration	MEFL + 3 dB 1 min	MEFL + 3 dB 2 min	MEFL + 3 dB 1 min	MEFL 1 min
Random Vibration (1) Level Duration	Low Frequency Random Vibration 1 minute	MEFL + 3 dB 2 min/axis	MEFL + 3 dB 1 min/axis	MEFL 1 min/axis
Sine Vibration Level Duration	No test	1.4 x FA level 2 octaves/min	1.4 x FA level 4 octaves/min	MEFL 4 octaves/min
Pyro Shock (2) Level or Firings	2 actual device firings	1.4 x MEFL (2 shocks/axis) or 2 firings	1.4 x MEFL (1 shock/axis) or 2 firings	Normally performed to PF levels
Thermal: Orbiter bus electronics (3)	Within AFT and not to exceed assembly PF	-35°C to +70°C or AFT-15°C to AFT+20°C [whichever is greater] Duration: 144 hrs hot; 24 hrs cold	-35°C to +70°C or AFT-15°C to AFT+20°C [whichever is greater] Duration: 144 hrs hot; 24 hrs cold	AFT-5°C to AFT+5°C Duration: 60 hrs hot; 8 hrs cold
Thermal: Orbiter mechanisms (3)	Within AFT and not to exceed assembly PF	AFT-15°C to AFT+20°C	AFT-15°C to AFT+20°C	AFT-5°C to AFT+5°C
Thermal: Payload electronics (3)	Within AFT and not to exceed assembly PF	-35°C to +70°C or AFT-15°C to AFT+20°C [whichever is greater] Duration: 144 hrs hot; 24 hrs cold	-35°C to +70°C or AFT-15°C to AFT+20°C [whichever is greater] Duration: 144 hrs hot; 24 hrs cold	AFT-5°C to AFT+5°C Duration: 60 hrs hot; 8 hrs cold
Thermal: Payload detectors, optics (3)	Within AFT and not to exceed assembly PF	AFT-15°C to AFT+20°C Duration: 144 hrs hot; 24 hrs cold	AFT-15°C to AFT+20°C Duration: 144 hrs hot; 24 hrs cold	AFT-5°C to AFT+5°C Duration: 60 hrs hot; 8 hrs cold
Total Ionizing Dose & Displacement Damage	RDF = 2			
Single Event Effects	RDF = 1 (applied to environments)			
EMC Susceptibility	Expected Levels + 6 dB (for pyrotechnic devices, +20 dB if by analysis)			
EMC Emissions	Expected Levels – 6 dB (for pyrotechnic devices, -20 dB if by analysis)			

Table A-1 Orbiter and Payload Environmental Design/Test Margin Requirements

RDF = Radiation Design Factor = Radiation Design Margin (RDM)
EMC = Electromagnetic Compatibility
MEFL = Maximum Expected Flight Level
FA = Flight Acceptance
AFT = Allowable Flight Temperature
PF = Protoflight

(1) = NASA-STD-7001, Payload Vibroacoustic Test Criteria
(2) = NASA-STD-7003, Pyroshock Test Criteria
(3) = Design, Verification/Validation and Operations Principles for Flight Systems, JPL D-17868, 6-15-99

Figure A-1 Qual/PF/FA/AFT Temperature Relationships



APPENDIX B

ENVIRONMENTAL PROGRAM DEFINITIONS

Acoustics – a test intended to simulate the actual acoustic environment occurring during launch vehicle ascent due to engine exhaust noise and boundary layer noise. The test is typically specified to be applied to systems and subsystems/assemblies that have relatively large surface area-to-mass ratios.

Allowable Flight Temperature (AFT) – The worst case temperature extremes, at the cold and the hot extreme, that flight hardware will be allowed to experience in its mission when all uncertainties of temperature control design, analyses, and temperature control performance are taken into account.

Assembly – An assembly is the lowest level of replaceable unit in the subsystem/system configuration. An assembly is a functional unit that is viewed as an entity for purposes of analysis, manufacturing, maintenance or record keeping. Examples are valves, electrical harnesses, individual electronic boxes such as transmitters, receivers or multiplexers and science Payload optics or detectors. Assemblies are normally environmentally tested. (see definitions for subsystem and system.) Some aerospace organizations use the term “Component”.

Component – ‘Component’ is the term some aerospace organizations use as equivalent to a JPL ‘assembly’ as defined above.

Design Temperature Limits
Temperature limits to which assemblies are designed to meet functional and performance specifications.

EMC Emissions – Tests performed to determine radiated electromagnetic emissions from a subsystem or assembly (measured at 1 meter) and to determine the interference or noise generated by an assembly or subsystem and conducted outside that unit on its power and/or signal lines.

EMC Isolation – Resistance measurements made from circuit active leads or common to the structure to verify that those circuits required to be isolated from the structure to satisfy grounding requirements are in fact isolated.

EMC Susceptibility – Tests to determine the susceptibility of a system/subsystem or assembly to electromagnetic radiation or the susceptibility of signal and power circuits to injected transients and signals that may be experienced during any phase of its intended life.

Environmental Design Requirements – The specific environments and limits of stresses or other environmental conditions which the flight hardware must be designed to withstand without exceeding allowable degradation or failure. Specific environments and levels are provided for each flight hardware classification.

Environmental Safety Compatibility/Assessment – An assessment used to verify that adequate margin exists in systems and subsystems exposed to ground, launch, and mission environments so they do not present any personnel or launch vehicle safety hazards.

Environmental Test and Analysis Requirements Matrix – a table or matrix in the Environmental Requirements document that specifies the environmental test and analysis program (i.e., defines the tests to be performed and defines the assembly and system configurations that will be environmentally tested). It also defines the environmental verification that will be achieved through analyses either independent of testing or in conjunction with testing.

Environmental Test Authorization and Summary (ETAS) – A form used to establish and document test readiness, test authorization, and test results.

Environmental Test Plan – A plan which defines the methods for implementing environmental testing of flight systems. A test plan normally includes the test approach, procedure, instrumentation requirements, test levels, and data monitoring and reduction requirements.

Environmental Test Procedures – Documents prepared to define the detailed implementation process for each assembly level environmental test required by the applicable test specifications.

ESD Tests – Tests performed using an electrostatic discharge (or spark) to simulate the effects of arc discharges that may occur between two adjacent Orbiter materials that are at different potentials due to differential charging in certain charged particle environments in space. These tests are conducted to determine the electromagnetic interference that may result when such discharges occur.

Flight Acceptance Tests

Flight Acceptance (FA) environmental tests are formal environmental tests performed on flight hardware and spares to verify flight workmanship quality, but only when a previous protoflight or qualification test has been performed on an identical item to qualify the design. FA environmental tests also may be used to verify the quality of reworked flight hardware. FA testing includes meeting functional specifications under flight acceptance environments.

Flight Acceptance testing should be evaluated for use on a case-by-case basis. If it is determined by a Heritage Review that previous qualification or protoflight test levels on a heritage assembly envelope those required for the MRO assembly, and the heritage design and operation is not modified in such a way as to negate the previous qualification, then the assembly may be Flight Acceptance tested.

General Assembly and System Test Specifications – The documents that define and control the specific environmental test conditions and other requirements applicable to environmental testing of flight Orbiter and Payloads.

Launch Pressure Profile – A test to simulate the reduction in air pressure as the payload experiences launch, ascent and mission transition. The design verification test shall be accomplished using a pressure decay rate margined above the profile expected to be experienced by the launch vehicle. Due to general facility constraints, a venting analysis is generally performed to fulfill this requirement.

Protoflight Tests

Protoflight (PF) tests are formal environmental tests performed on flight hardware, which is intended to be flown, and which has no previous qualification test heritage. Protoflight testing accomplishes in one test the combined purposes of design qualification and flight.

Protoflight thermal test levels and durations are identical to qualification test levels and durations. Protoflight dynamics test levels are equivalent to qualification test levels; however, the duration is lowered to flight acceptance duration. PF testing includes meeting functional specifications under protoflight environments.

Pyro Shock – A test intended to simulate the dynamic effects resulting from the firings of pyrotechnic devices on the Orbiter, etc. Implementation methods include the actual firings of pyro devices, shock produced by a vibration shaker and by impacts with a hammer or a swing.

Qualification Tests

Qualification tests are formal environmental tests performed on a dedicated Qualification Model or flight-like Engineering Model of flight which is not intended to fly in order to demonstrate flight design adequacy and quality workmanship. Qualification thermal environmental testing is equal to Protoflight testing. Qualification dynamics environmental testing is equal to Protoflight level testing in magnitude and exceeds it in duration. Qualification testing implies meeting all functional specifications in the operating environments.

Radiation Design Factor (or Margin) – The ratio of electronic part radiation capability to the local part radiation environment.

Random Vibration – A vibration test intended to simulate the acoustic vibration mechanically transmitted to the Orbiter system or subsystem/assembly through its attachments. Large spacecraft which are more severely excited by direct acoustics may not be required to be subjected to random vibration tests.

Solar, Albedo and Planetary IR Intensity – Simulation of the combined effects of direct solar illumination, planetary solar reflections (albedo) and planetary infrared (IR) emissions during system level thermal vacuum testing. This may be accomplished by true solar simulators, infrared panels, chamber wall temperature control or combinations of these. The Orbiter thermal control design concept (coatings, insulation, louvers, etc.) will determine which simulation techniques are appropriate.

Subassembly – The term subassembly denotes two or more parts joined together to form a stockable unit which is capable of disassembly or part replacement. Examples are a printed circuit board with parts mounted or a gear train.

Subsystem – A subsystem is a combination of two or more assemblies and any interconnecting cables or tubing. A subsystem is composed of functionally related assemblies that perform one or more prescribed functions. Typical subsystems are electric power, attitude control, telemetry, instrumentation, command and data handling, structure, thermal control and propulsion. Subsystems relate primarily to Orbiter hardware.

System – A system is the composition of interdependent hardware subsystems and/or assemblies capable of performing or supporting an operational role. A system includes all operational equipment and software. Examples of systems include Orbiter, Payloads delivered to a sponsor for integration into Orbiter or Payloads to be attached to the Shuttle or Space Station.

Thermal Cycling – A test performed (in vacuum or at atmospheric pressure) on assemblies which experience significant thermal cycling during the intended mission or used as a workmanship test. Stabilization criteria for high and low temperature plateaus are a function of the response of individual assemblies. The minimum number of cycles is based on workmanship considerations and the maximum number is mission dependent.

Thermal Shock – A test performed (in vacuum) on assemblies which experience wide ranges of temperature excursion with a high rate of temperature change (solar panels, etc.). Stabilization criteria are a function of the response of individual assemblies. Transition rates are simulated flight rates since higher rates are normally not attainable in vacuum. The minimum number of cycles is based on workmanship considerations and the maximum number is mission dependent.

Thermal Vacuum Duration/Pressure – The time elapsed under which the system and/or subsystem/assembly is subjected to, stabilized at and operating within specification while immersed in a vacuum. Test time required to achieve vacuum conditions and transition time between temperature levels are not included in the requirements in the assembly level test specification and must be added to determine the total test time needed to conduct thermal vacuum tests.

Thermal Vacuum Temperature – The high and low temperature levels at which the system and/or subsystem/assembly will be stabilized and required to be operating within the design performance specifications, during thermal vacuum tests.

APPENDIX C

EMC DESIGN BEST PRACTICES FOR SPACECRAFT

Structure Shielding

The Orbiter structure shall form an all-enclosing electromagnetic shield offering at least 40 dB attenuation to electromagnetic fields in the frequency range from 10 kHz to 0 GHz. This is commonly called a Faraday Cage construction.

This shall also include special attention to (but not limited to) penetrating aperture holes, seams and joints, edges, corners, feedthroughs, etc. Examples of such holes are solar array shafts, wiring from solar arrays to power converters, RF coaxial cables and waveguides, external cables and wires such as temperature transducers, squib wiring, observational Payload apertures, etc.

Thermal blanket material may not be considered part of the Faraday Cage.

Payloads not contained inside the main body of the Faraday Cage shall provide their own individual Faraday Cage shielding to prevent interference from the radar RF fields, and to prevent RF leakage that would affect the radar receiver.

Cable Shielding

All cables on the exterior of the satellite's Faraday Cage shall have shielding offering at least 40 dB of attenuation to electric fields over the frequency range of 10 kHz to 10 GHz. This may be accomplished by high quality double layered braid shield, or braid shield with an overwrap of aluminum or other conductive metal foil. The cable shield shall be terminated at the Orbiter or box's Faraday Cage with a 360 degree EMI backshell, at both ends of the cable. There shall be no unshielded part of any cable bundle, and there shall be no pigtail termination of any cable shield.

This cable overshield is separate and different from any shield contained in the cable, such as a twisted shielded wire pair; these latter shields may be carried in the cable and may be terminated separately through pins in the connectors according to the system grounding and shielding plan.

Grounding and Bonding and Electrical Interfacing

The Orbiter system integrators should require a design that has good ground planes built into the Orbiter (for example, the walls of the Faraday Cage) with low impedance bonding across all ground reference regions. The Orbiter system integrators should also require good electrical grounding architecture, providing short ground paths from subsystem electronics packages to the ground planes. There should be no deliberate current in the structure or ground, and there should be no ground loops permitted between subsystems. This requires system integrator control of interfaces between subsystems, carefully specifying isolation at subsystem interfaces, both for power and signal/data. NASA-HDBK-4001 (Feb. 17, 1998) describes such good grounding practice.